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(NASA-CR-157881) EFFECTS OF IGNITER AND MOTOR CHARACTERISTICS ON THE IGNITION TRANSIENT OF A SOLID ROCKET ENGINE: COMPUTER PREDICTIONS AND TEST FIRINGS (Princeton Univ., N. J.) 22 p

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PRINCETON UNIVERSITY

DEPARTMENT OF

AEROSPACE AND MECHANICAL SCIENCES

EFFECTS OF IGNITER AND MOTOR CHARACTERISTICS ON THE IGNITION TRANSIENT OF A SOLID ROCKET ENGINE: COMPUTER PREDICTIONS & TEST FIRINGS

Aerospace & Mechanical Sciences Report No. 809

by

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October 1967

This research was sponsored by the Office of Advanced Research and Technology, National Aeronautics and Space Administration, under Research Grant NsG 200-60, Supplement No. 5.

Transmitted by

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This report is a preprint of a paper presented at the Fourth ICRPG Combustion Conference, Menlo Park, California, October 9-11, 1967. It is scheduled to appear in the Proceedings of that Conference, which will be distributed on the CPIA mailing list. Additional copies of this preprint are obtainable from Princeton University.

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SPONSORSHIP

This research was sponsored by the Office of Advanced Research and Technology, National Aeronautics and Space Administration under Research Grant NsG 200-60, Supplement No. 5. The Program Manager for this program was Mr. Robert W. Ziem, Chief, Solid Propellant Technology, Office of Advanced Research and Technology. The Technical Manager for this program was Mr. Warren L. Dowler, Solid Propellant Engineering Section, Jet Propulsion Laboratory. The Contracting Officer for the program was Mr. T. L. K. Smull, Office of Grants and Research Contracts.

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ABSTRACT

The research reported in this paper is aimed at the development of a method of predicting the thrust-time curve during the ignition transient of a solid propellant rocket engine. The underlying theory, which has been reported in previous publications, comprises a treatment of the three important phases of the ignition transient, namely, the ignition lag, the flame spreading process and the final chamber filling process.

Several individual projects are reported in this paper. One is a systematic computer study of the effects of engine design variables on the resulting ignition transient. The second comprises a series of engine firings conducted to test the validity of the computer predictions, and good agreement is reported for the various comparisons. A third project was aimed at testing the prediction analysis on rocket engines with practical star and cylinder grain geometries, and here again the comparisons between computer-predictions and firing traces were deemed satisfactory. Finally, a brief progress report is given on a project involving the ignition transient of a service type gas generator with a magnesium-teflon igniter.

It appears, from the results accomplished so far, that the computation method that has been developed is generally successful for a certain class of motors ignited with pyrogen igniters. Future work will be concentrated on extending these principles to other classes of rocket engines.

Introduction: Theoretical Model

In the past few years a number of studies have been reported on the combustion and gas dynamic processes in a solid propellant rocket engine during the ignition transient. Research attention has been concentrated on the mechanism of flame spreading over the surface of a solid propellant grain and on the related processes of flame initiation and gas dynamic chamber filling[1-14]. As a result of several continuing research programs in this laboratory, an analytical model has been developed to predict the entire ignition transient for a particular class of rocket engines. For the flame spreading part, the theory rests on the concept that ignition of each element of surface of the propellant grain occurs at the moment it attains a critical ignition temperature and that the heat causing ignition is propagated downstream by gas phase heat convection and not by conduction through the solid, which is the usual mode of flame propagation. The equations of the gas dynamics of the combustion chamber and the burning rate behavior of the propellant are written in nondimensional form. The system of equations is completed with an empirical equation characterizing the measured heat transfer between the flowing igniter gas and the propellant grain. The early development of this model is described in References 15-19 and the extension to its present form in References 20 and 21.

The theory in its present form most closely models rocket engines employing single axial perforation grains with large port-to-throat area ratios and which are ignited by gas-producing igniters located in the forward end of the rocket engine. However, the theory is capable of dealing with certain portions of the thrust transient for a much wider range of rocket engines, and ways can be seen for extending it eventually to cover the entire transient.

Verification of Model: 2-D Motor Firings Compared with Computer Predictions

Parallel to the theoretical development, an extensive experimental program was carried out to test the validity of the model. The major part of this program consisted of firings of a two-dimensional motor which employed a gaseous igniter located at the forward end of the grain. The details of this motor are given in Figure 1. The propellant was an unaluminized AP-PBAA composition. The equilibrium chamber pressures ranged up to 500 psia. The overall ignition transient from first application of the igniter until equilibrium operating conditions are reached was typically 200 msec in duration. These experiments included the measurement of igniter and motor pressure with fast response gauges, high speed photographic observation of flame initiation and flame spreading, and the measurement of heat transfer rates at various parts of the grain. The results to date have verified the main features of the physical model.

The preliminary results suggested the need for further experimental investigation of the heat transfer between the flowing igniter and combustion gases and the propellant surface. The refinements made to date are presented in Figure 1. Although these results can not be considered the ultimate description of the heat transfer, they appear to be sufficiently accurate to permit reasonable agreement between the theoretical predictions and the experimental test firings, as will be seen below.

The experimental test firings included in this paper consist of three series. In each series a single experimental parameter was systematically varied with all other parameters held constant. In Series A the exhaust nozzle was varied, in Series B the igniter duration was varied, and in Series C the igniter mass flow rate was varied with the total igniter mass held constant. These series are summarized in Table I.

The pressure transients for Series A are shown in Figure 2. The agreement between the theoretical predictions and the experimental test firings is illustrated in Figures 3 and 4. Figure 3 is typical of the good agreements that have been obtained and Figure 4 represents the greatest lack of agreement seen so far. The agreement between theory and experiment for all of the test firings in Series A and B was within these limits.

The interesting results of Series A and B are discussed below.

It can be easily shown that for the heat transfer correlation used ($Nu_x \sim Re_x^{0.8}$), the rise in surface temperature of the propellant is proportional to $(\dot{m}_{ign})^{0.3}$, for an igniter of constant total mass. Thus it can be seen that, although a given igniter may contain enough total mass for a successful ignition transient, if it is fired at too low a mass rate of flow, a misfire or a hangfire may occur. Series C (Figure 6) demonstrates that interesting type of hangfire.

Series C contains also an interesting pressure overshoot. The pressure overshoots seen in Series A and B (Figures 2 and 5) and Firing C-1 (Figure 6) are due to extended igniter durations. The overshoot observed in Firing C-4 (Figure 6) is due to the preheating in depth of the unignited portion of the propellant during the long delay between the start of igniter firing and the final rapid phase of flame spreading. The thermal wave in this case has penetrated far enough into the solid so that there is a layer of propellant which burns with an increased rate due to the higher initial temperature of the propellant. The increased burning rate of this thin layer results in a pressure overshoot.

Series C also demonstrates at least one limit of the theoretical model. As is seen in Figure 6 suitable predictions can be made for the more "normal" runs like C-1. However, for increasingly marginal ignition situations the quality of the predictions degenerates although the correct trends are predicted as seen for Firing C-2. This can be attributed to several factors not included in the present model. The simple ignition criterion used in the analysis is that the propellant is completely inert until a critical ignition temperature is suddenly reached. This ignores the details of the ignition process that have been learned through extensive work in this laboratory and elsewhere. In particular, it ignores the possible contribution of exothermic surface decomposition taking place before the effective ignition temperature is reached. This would convert a slow flame spreading process to a rapid one. This aspect of the problem requires further study to identify the active process, either surface heat generation or something else.

Application of Prediction Method to Practical Motor Configurations

Experimental test firings were carried out also for several rocket engines of more practical configuration than the two-dimensional motor. The first set of experiments involved laboratory-size rocket motors with a solid propellant igniter located at the forward end of the motor [25]. The igniter used a CMDB type propellant. The motor grains were an AP-PBAA unaluminized composition. The propellant charges weighed up to 0.33 lbs. Various hollow cylinder and star shaped grain designs were fired. Companion predictions were made for these test firings, using the same equations and heat transfer correlation developed for the two-dimensional motor described above. The quality of the predictions can be judged in Figure 7, which compares the experimental and theoretical curves for a star grain motor. Although there is good agreement between the theoretically predicted and experimental trends, the elimination of the consistent differences observed would require detailed study of the heat transfer along the points of the star and corrections for other three-dimensional effects. This has not been done.

The second set of experiments with practical configurations involves a rocket engine developed by the Frankford Arsenal. This engine has a hollow cylindrical grain of a nitrocellulose class propellant tapered at both ends to produce neutral burning. The charge weighs 2.8 lbs. and the firing duration is approximately 9 sec. It uses an igniter of magnesium and teflon which acts by direct particle impingement on the grain to achieve ignition. This is obviously different from the present theoretical model.

However, it is anticipated that the interval between igniter initiation and full ignitedness of the propellant surface will be small and that useful predictions of the subsequent dynamic pressure rise can be made.

Finally, the theory has been tested on the 120-inch rocket motor used on the Titan III-C Booster. The ignition transient of the 120-inch motor (five segments) has been published (Ref. 26), and enough information is available on the engine design to permit approximate calculations of the ignition transient on the basis of the present theory. Despite a large number of uncertainties in the available information and important departures from the present model, there is reasonable agreement between the predicted and measured transients.

Parametric Computer Study of Ignition System Design

The experimental program described above has demonstrated the general validity of the theoretical model and its ability to predict with reasonable accuracy the behavior of the ignition transient for normal situations and at least the correct trends for marginal cases leading to hangfires and misfires. This evidence lends credence to the computer study of igniter design given below. By systematic variation of igniter and motor parameters, the phenomena of pressure overshoot, ignition shock and several types of hangfires and misfires have been explored and characterized.

Before considering this series of calculations it will be beneficial to examine certain characteristics common to all the calculations. As seen in Figure 8 the initial conditions at the beginning of the igniter firing are

P_c= 15 psia, the ambient pressure, and T_g=2600°K, the igniter gas temperature. The pressure begins to rise due to the square wave onset of the igniter flow rate. The gas temperature rises due to the compression, but as the pressure reaches its pre-ignition equilibrium value, the temperature returns to its initial value. As the forward part of the grain becomes ignited, the presure begins to rise and the temperature decreases as the cooler combustion gas of the propellant mixes with the igniter gas.

When 30% of the grain has been ignited, the igniter is cut off, as planned for the series, causing a discontinuous change in the mass flow rate. The heat transfer to the propellant surface and the pressure in the chamber thereupon decrease momentarily. This causes a discontinuous change in the flame spreading rate. However, as the flame spreading continues the presure rises sharply again. The chamber gas temperature rises simultaneously due to the compression, but the rise is slowed by the addition of the cooler propellant combustion gases.

When flame spreading is completed, a process of feedback ensues whereby the pressure increases,- thus increasing the burning rate, and thereby sending more mass into the chamber to further increase the pressure. In this manner equilibrium conditions are reached in the chamber.

Six sets of thrust transients have been analyzed in this computer study. The six sets represent: (1) a series with various igniter flow rates ranging from 2.1×10^{-3} to 107×10^{-3} lbs/sec/in^2, Figure 9; (2) a series with various rocket nozzle throat areas, Figure 10; (3) a series with various port cross-section areas, Figure 11; (4) a series with excessive igniter durations to demonstrate pressure overshoots, Figure 12; (5) a series with abnormally low igniter burning rates to demonstrate the possibilities of hangfires, Figure 13; (6) a series with calibrated frangible diaphragm closures for the main motor exhaust nozzle to test their efficacy in producing prompt thrust rises, Figure 14.

A few of the more interesting results of the computer study and of the experimental firings displayed earlier can be summarized as follows. Increasing the igniter mass flow rate decreases the induction time to first ignition and increases the rate of flame spreading. This is due to the increased heat flux to the propellant surface. The induction interval is relatively independent of the magnitude of the exhaust nozzle area, if other things are held fixed. Decreasing the port cross-sectional area decreases the time to reach equilibrium operating conditions, i.e., steepens the rate of rise. The maximum pressure overshoot above the final equilibrium value increases with increasing igniter mass flow rate.

The maximum rate of rise of pressure in the chamber is found to occur at the beginning of the chamber filling interval, but it depends also on the chamber pressure at the end of flame spreading and hence on the rate of flame spreading. In particular, the maximum rate of rise of pressure (which occurs after flame spreading is complete) can be reduced by arranging conditions to slow down the rate of flame spreading. Increasing the igniter flow rate, decreasing the port cross-sectional area and decreasing the nozzle area all increase the maximum rate of rise of pressure.

It was found, as expected, that a closure of the exhaust nozzle that ruptures at some mid-pressure hastens the rise to full pressure. However, it was observed that the effect was much smaller than is popularly supposed in the rocket design field.

Conclusions; Problems for Future Work

In conclusion, it seems that the mechanism of flame spreading for a single perforation rocket grain ignited at the forward end is well enough understood to produce useful design predictions of thrust transients. It is now appropriate to tackle more difficult (and more practical) classes of design: small port-to-throat area ratios (high volumetric loading); gas-less igniters (so-called) or igniters with intense radiation; multi-perforated grains; aft end ignition; complex grain perforations and flow channels; etc. Such investigations are for the future.

ACKNOWLEDGEMENT

A number of workers in the Guggenheim Laboratories at Princeton University contributed to this work. Mr. C. R. Felsheim carried out the processing of the solid propellant used in the program and collaborated with the authors in the development of the experimental equipment. Advice on matters of instrumentation and assistance with the experimental firing program was given by Mr. S. O. Morris. Credit for the photographic work and related areas goes to Mr. E. R. Crosby. Mr. J. H. Semler was instrumental in the fabrication of the experimental equipment.

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TABLE I

IGNITION TRANSIENT FIRING EXPERIMENTS

Object: To compare computer predictions of $P(t)$ with firing traces.

Series A: Fixed Igniter Flow ($\dot{m}_{ign} = 18 \times 10^{-3}$ lbm/sec), Fixed Igniter Duration (140 msec), Exhaust Nozzle Systematically Varied.

A-1: $d_t = .170$ inches
A-2: $= .189$ inches
A-3: $= .219$ inches

Series B: Fixed Igniter Flow ($\dot{m}_{ign} = 18 \times 10^{-3}$ lbm/sec), Fixed Exhaust Nozzle ($d_t = .189$ in.), Igniter Duration Systematically Varied.
t_{ign}=time between opening and closing of igniter flow valves.

B-1: $t_{ign} = 140$ msec
B-2: $= 116$ msec
B-3: $= 100$ msec

Series C: Fixed Exhaust Nozzle ($d_t = .189$ in.)
Fixed Total Igniter Mass $(m_{ign})_{TOT} = 1.44 \times 10^{-3}$ lbm,
Igniter Flow Systematically Varied.

C-1: $\dot{m}_{ign} = 18.00 \times 10^{-3}$ lbm/sec
C-2: $= 13.41 \times 10^{-3}$ lbm/sec
C-3: $= 9.15 \times 10^{-3}$ lbm/sec
C-4: $= 4.89 \times 10^{-3}$ lbm/sec

TABLE II

TABLE OF SYMBOLS FOR ILLUSTRATIONS

d_t = diameter of exhaust nozzle.

$\dot{m}_{ign}$ = igniter mass flow rate.

P = nondimensional chamber pressure

$= \frac{\text{instantaneous chamber pressure}}{\text{equilibrium chamber pressure}} = \frac{P_c}{P_{eg}}$

S = nondimensional area burning

$= \frac{\text{instantaneous burning area}}{\text{total available burning area}}$

T = nondimensional chamber temperature

$= \frac{\text{instantaneous chamber temperature}}{\text{propellant adiabatic flame temperature}}$

τ = nondimensional time

$= \frac{\text{time}}{\text{t*, the characteristic residence time}}$

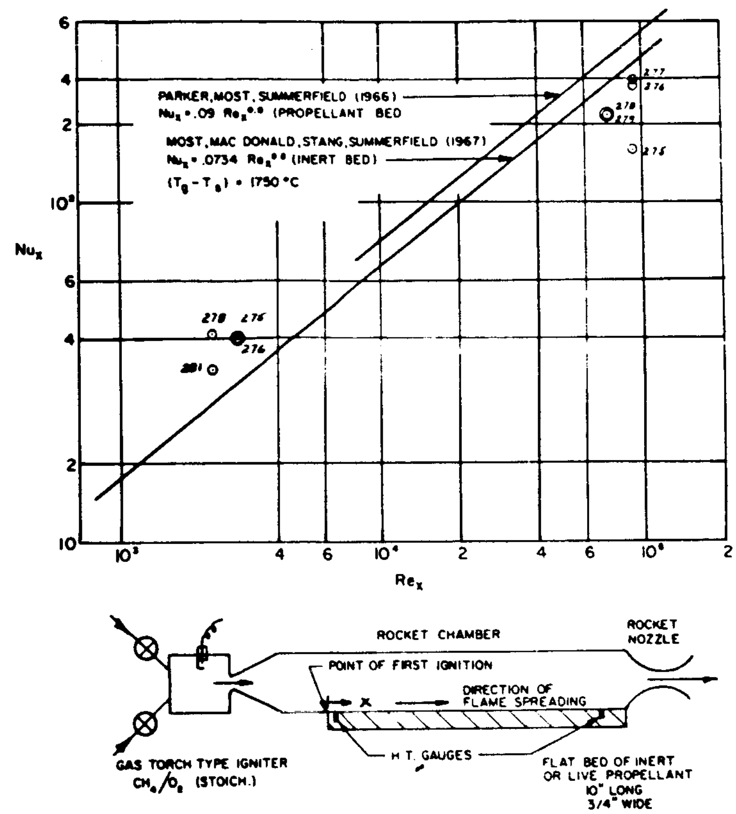


Figure 1 Schematic Drawing of Experimental Rocket Motor and Experimental Heat Transfer Correlations

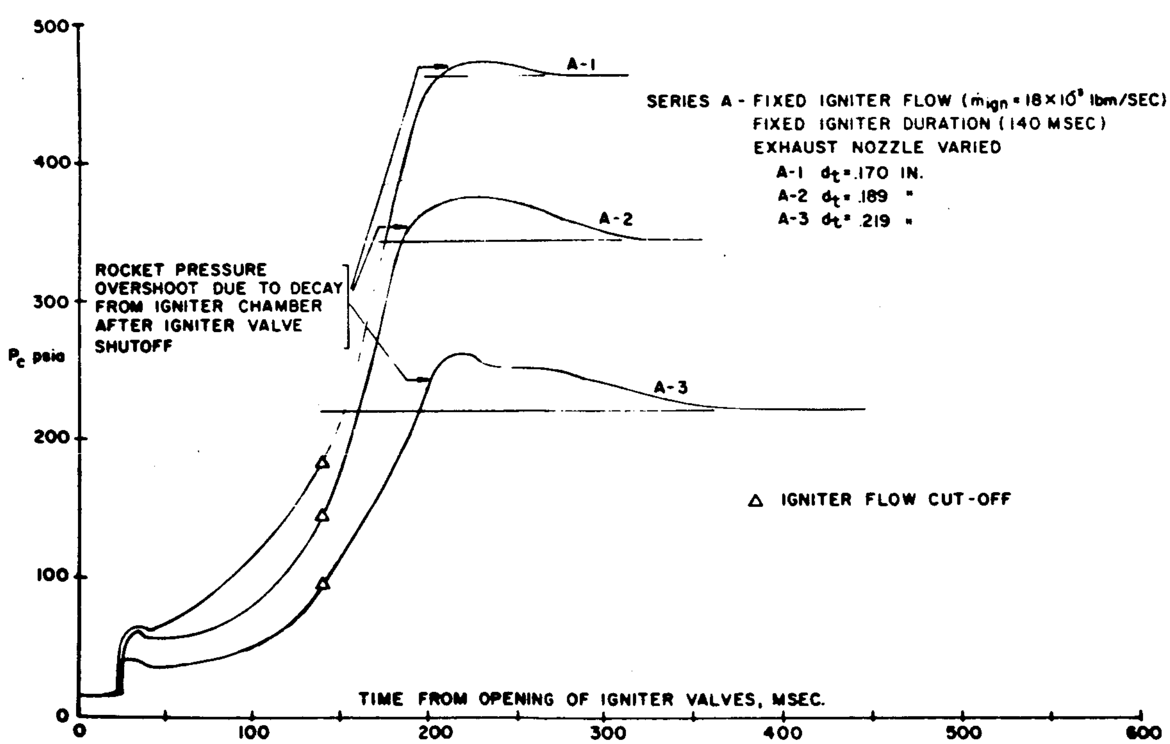


Figure 2 Series A - Effect of Varying Exhaust Nozzle Area

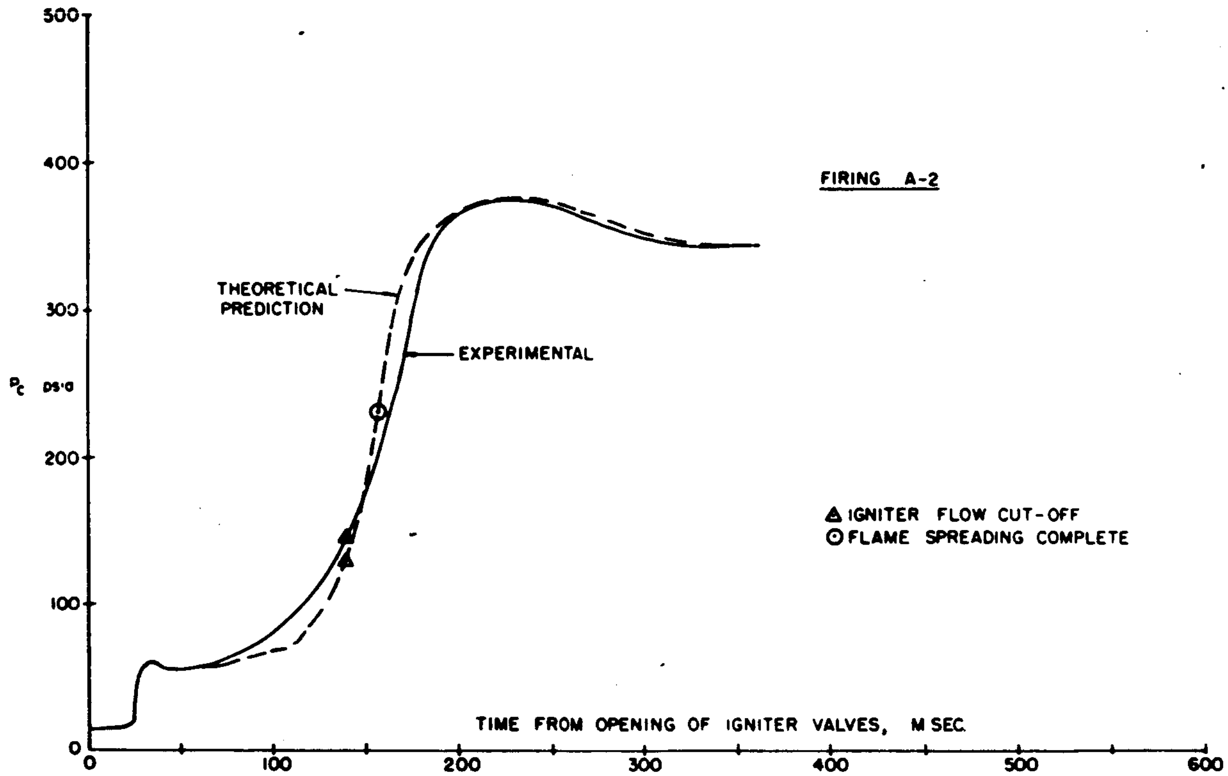


Figure 3 Typical Good Agreement Between Theory and Experiment

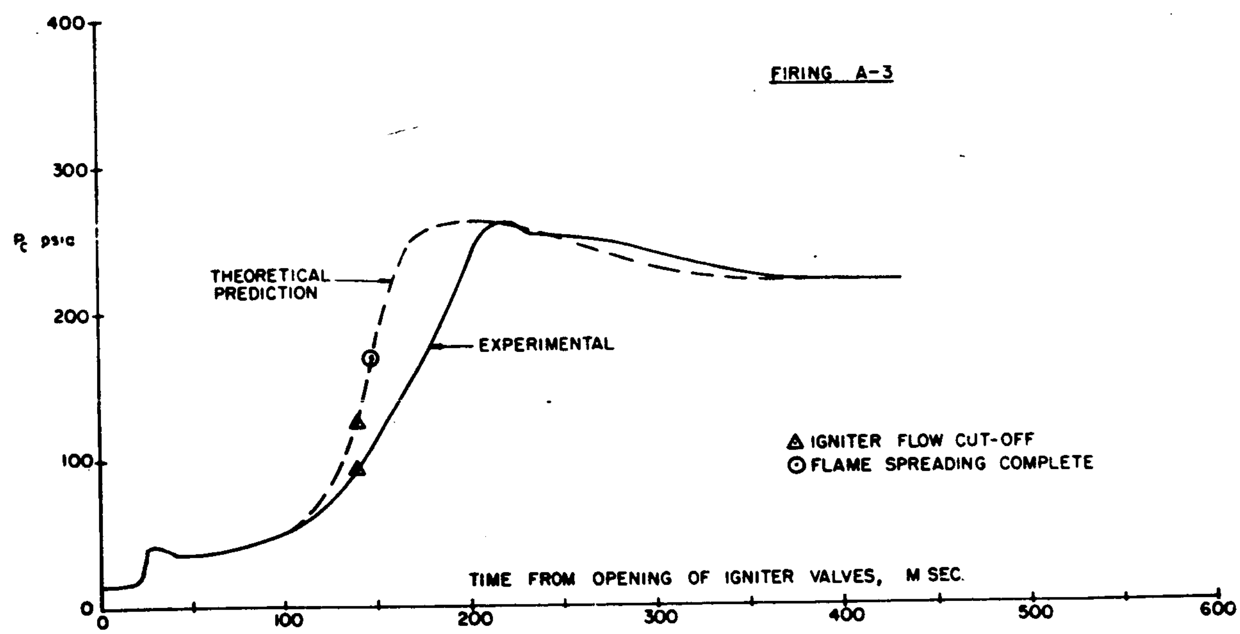


Figure 4 Typical Greatest Lack of Agreement Between Theory and Experiment

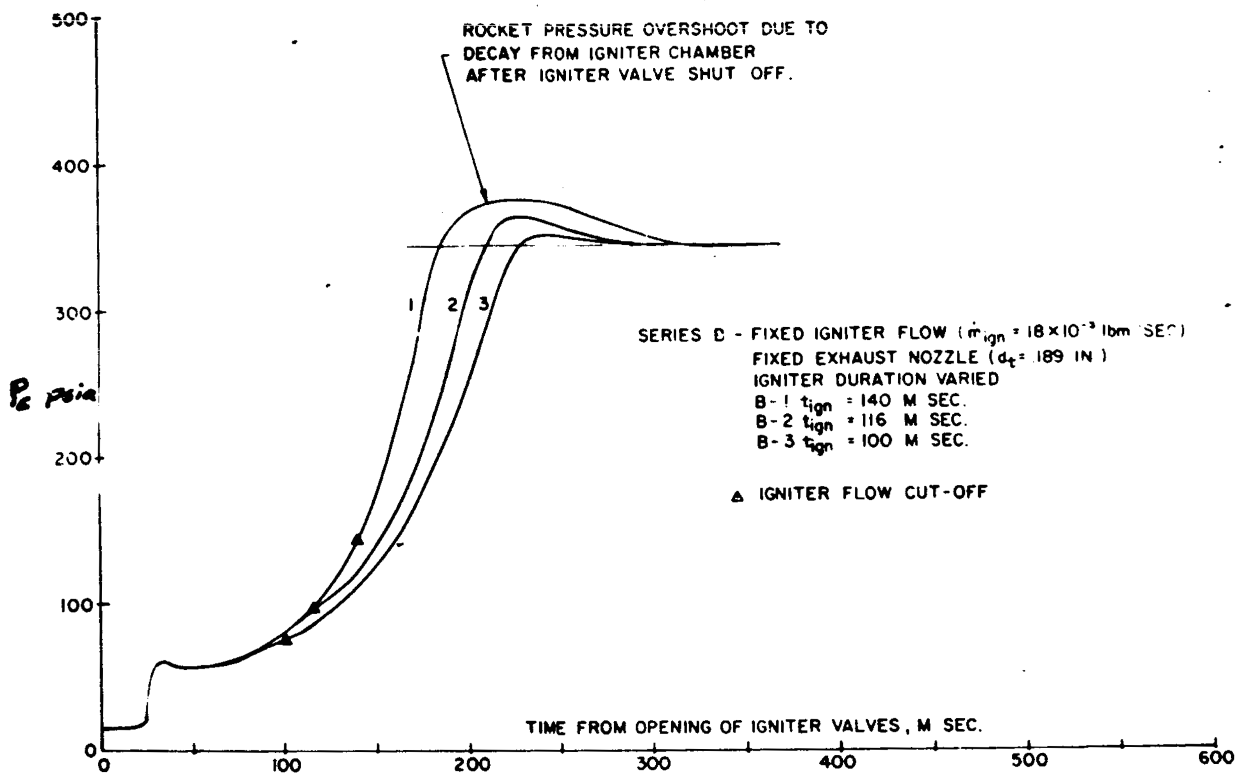


Figure 5 Series B - Effect of Varying Igniter Duration

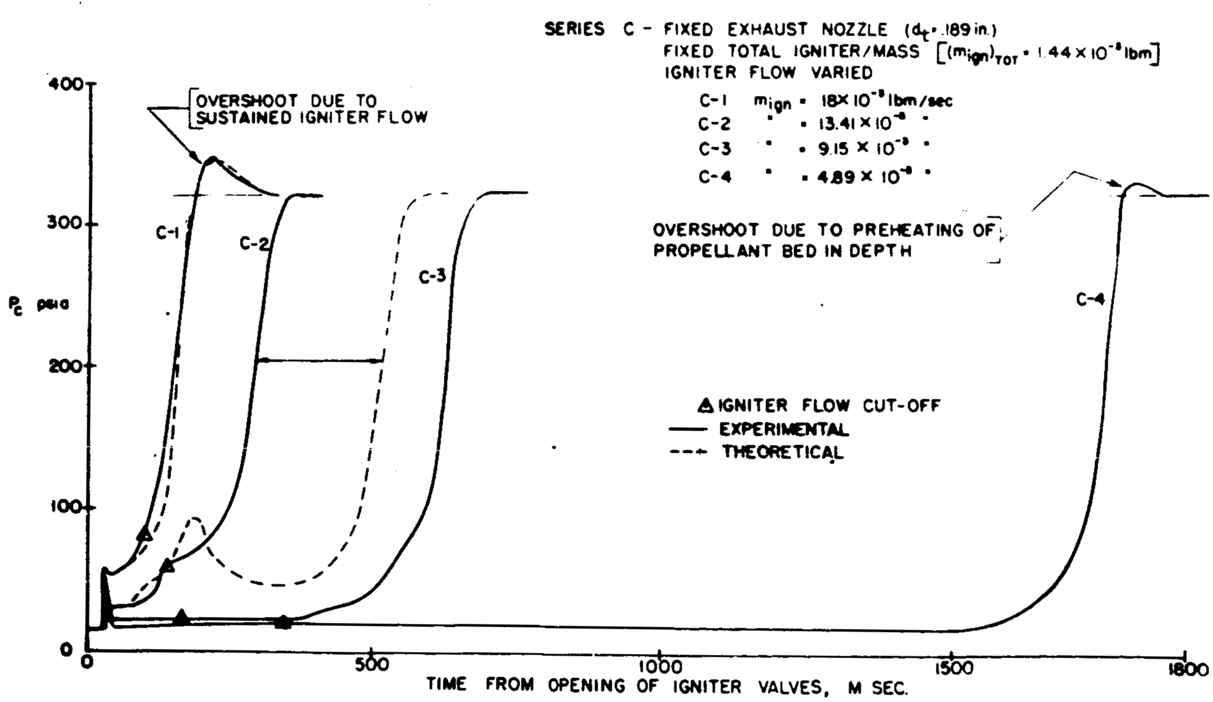


Figure 6 Series C - Effect of Varying Igniter Flow Rate with the Total Igniter Mass Constant

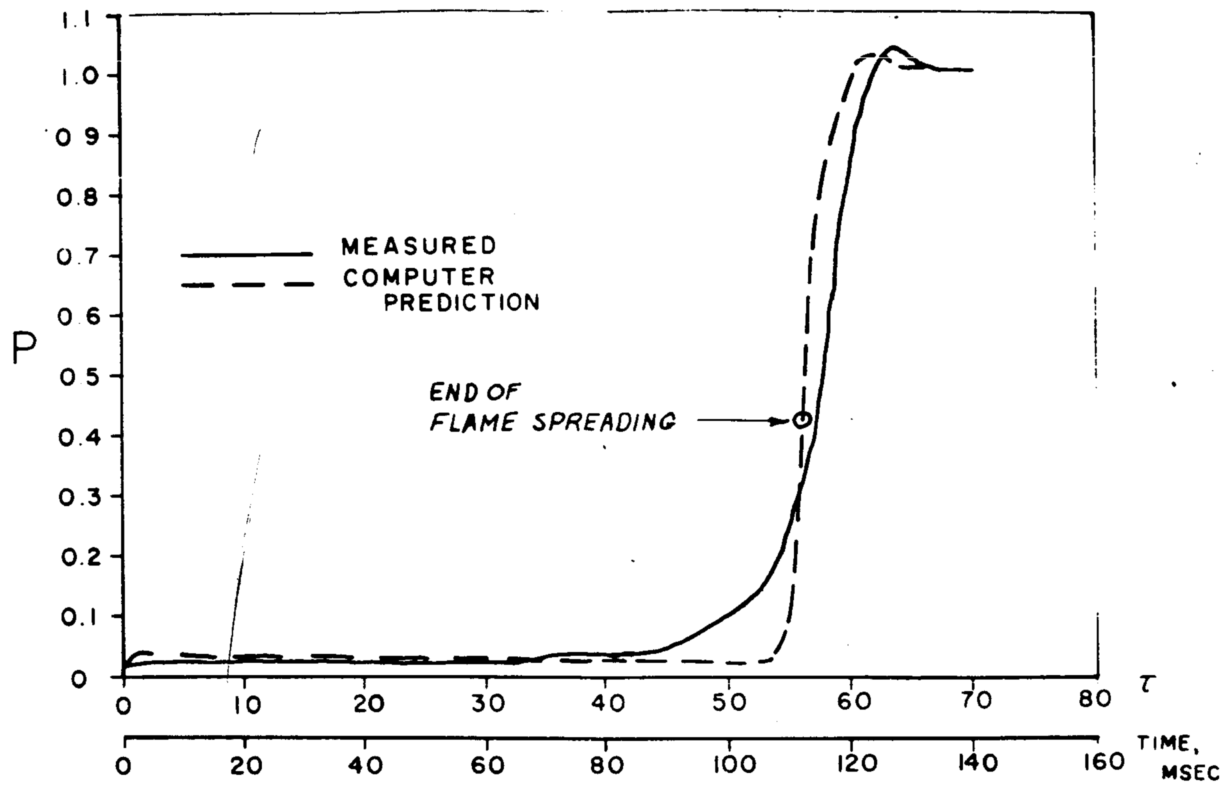


Figure 7 Comparision of Theory and Experiment for a Typical Star Grain Rocket Motor

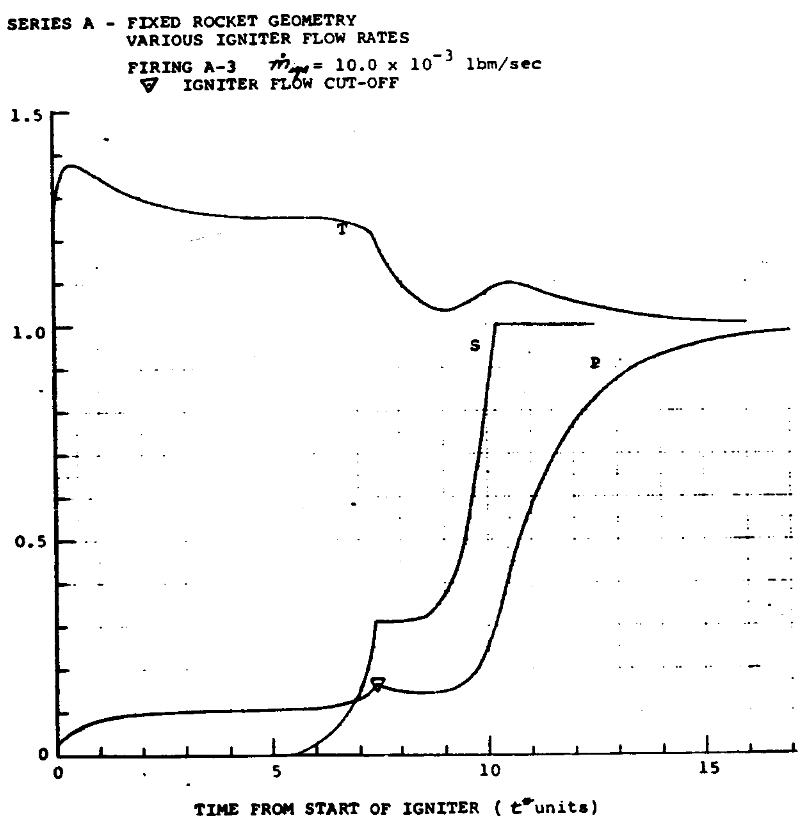
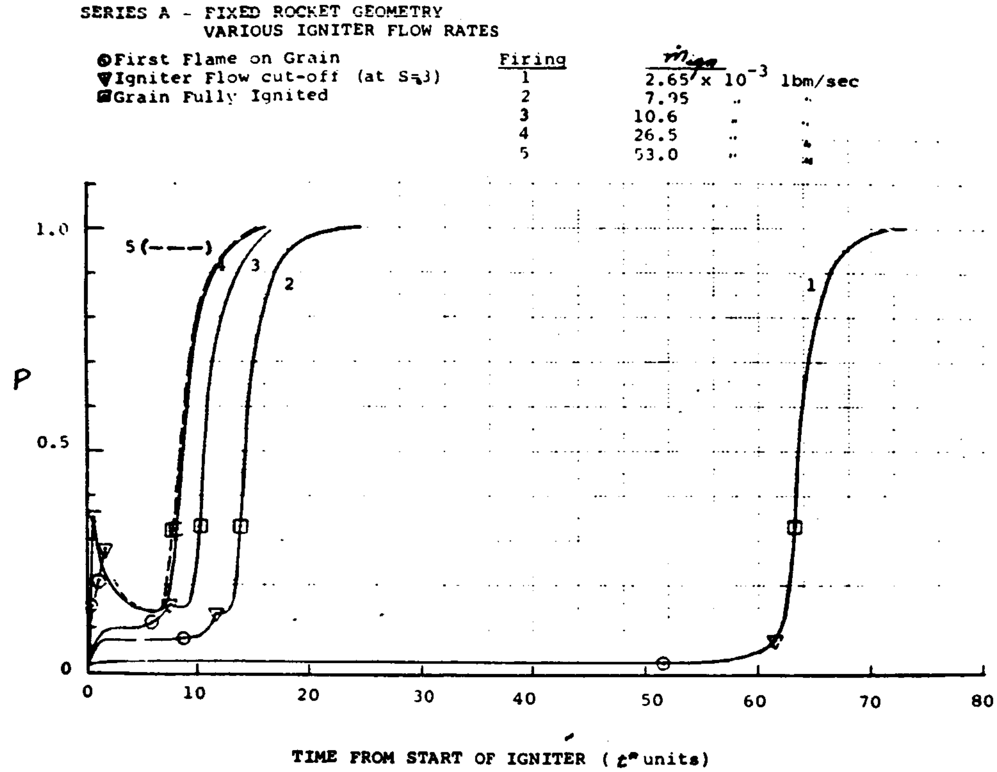
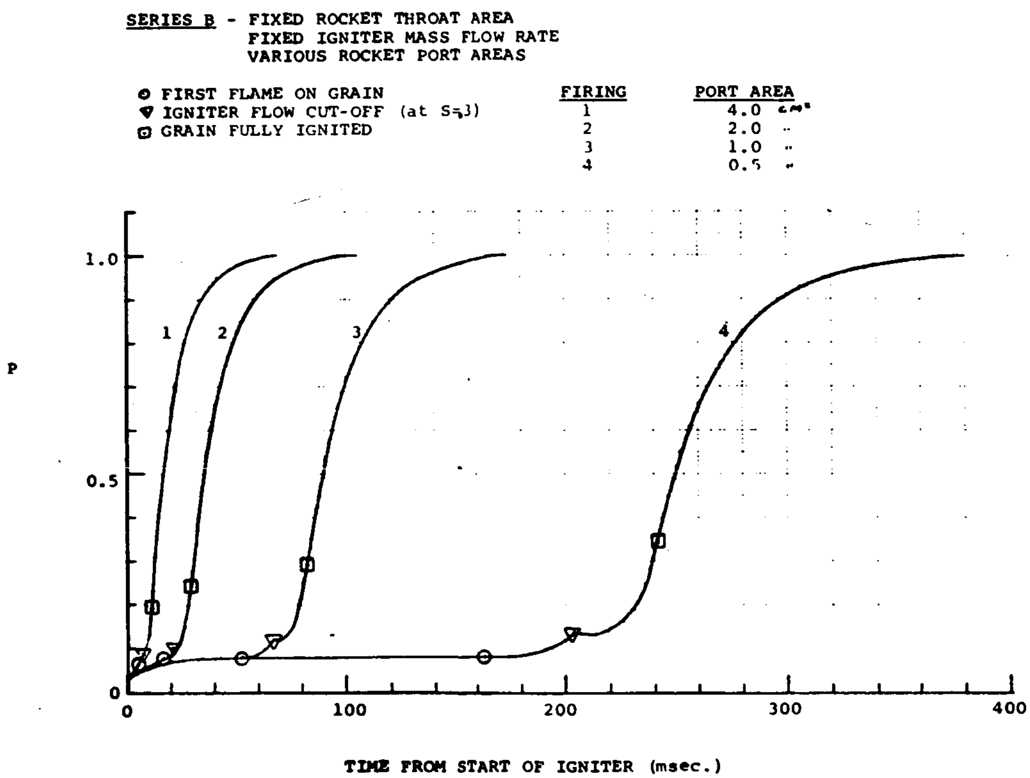


Figure 8 A Typical Theoretical Prediction



<u>Figure 9</u> Theoretically Predicted Effect of Varying Igniter Flow Rate



<u>Figure 10</u> Theoretically Predicted Effect of Varying Port Area

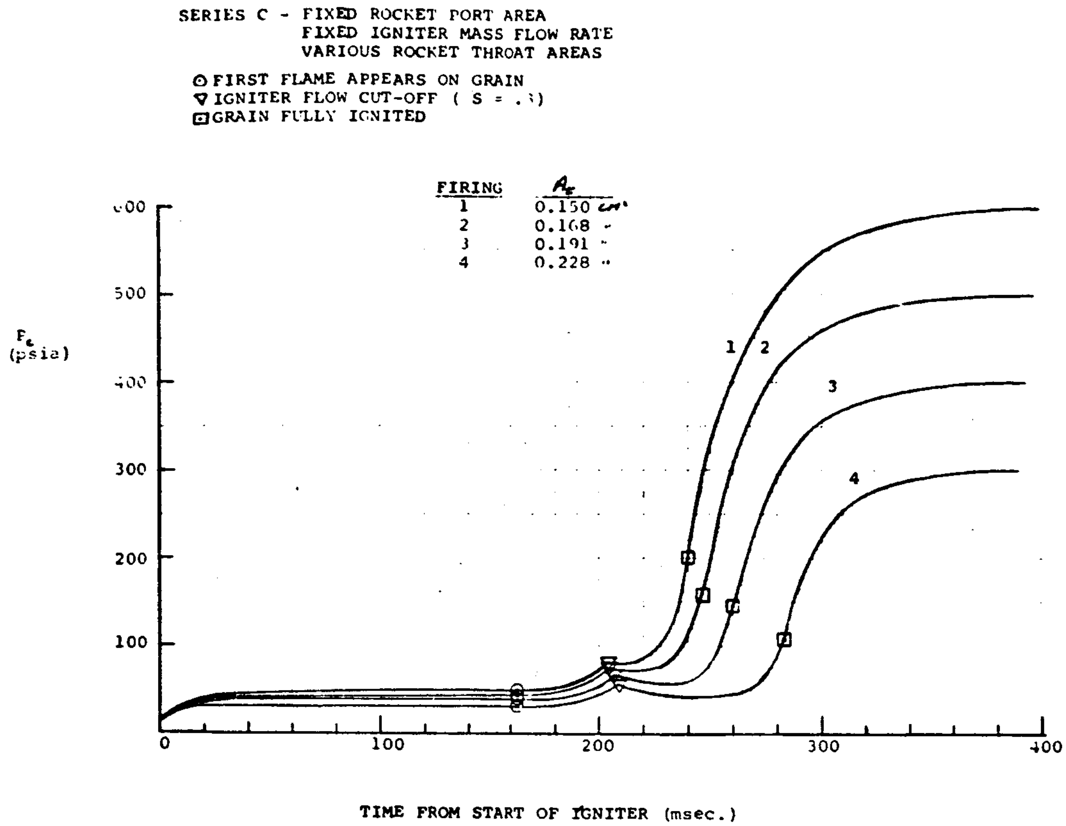


Figure 11 Theoretically Predicted Effect of Varying Exhaust Nozzle Area

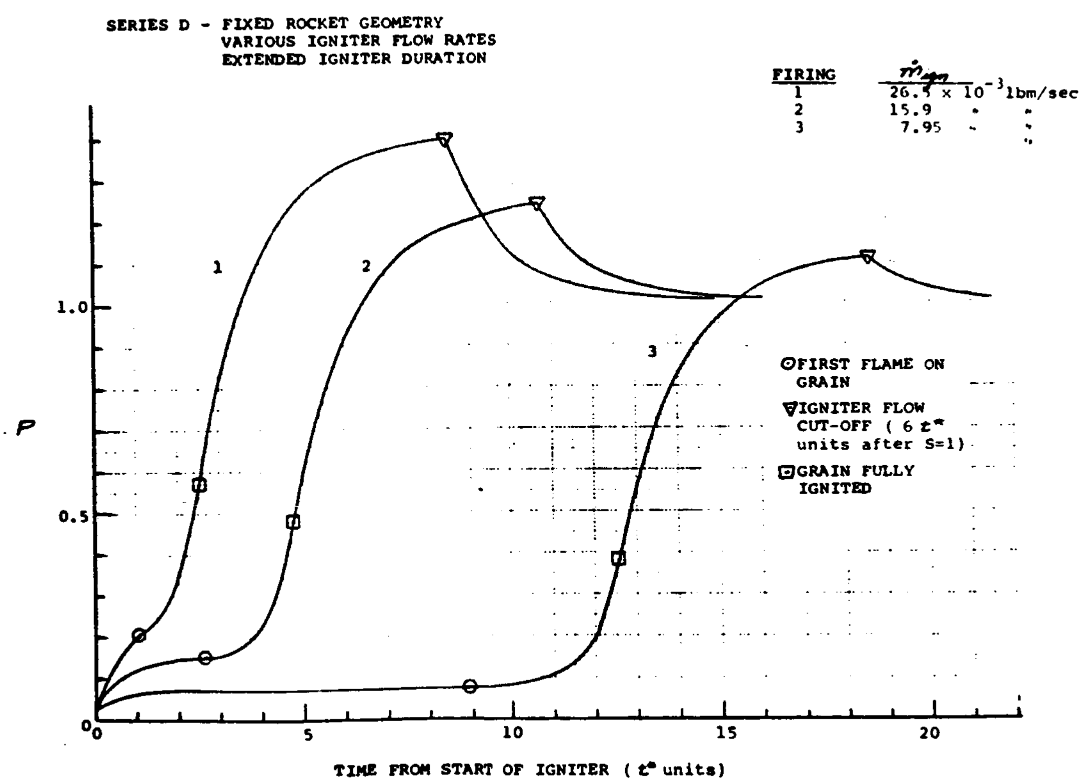


Figure 12 Theoretically Predicted Effect of Varying Igniter Duration

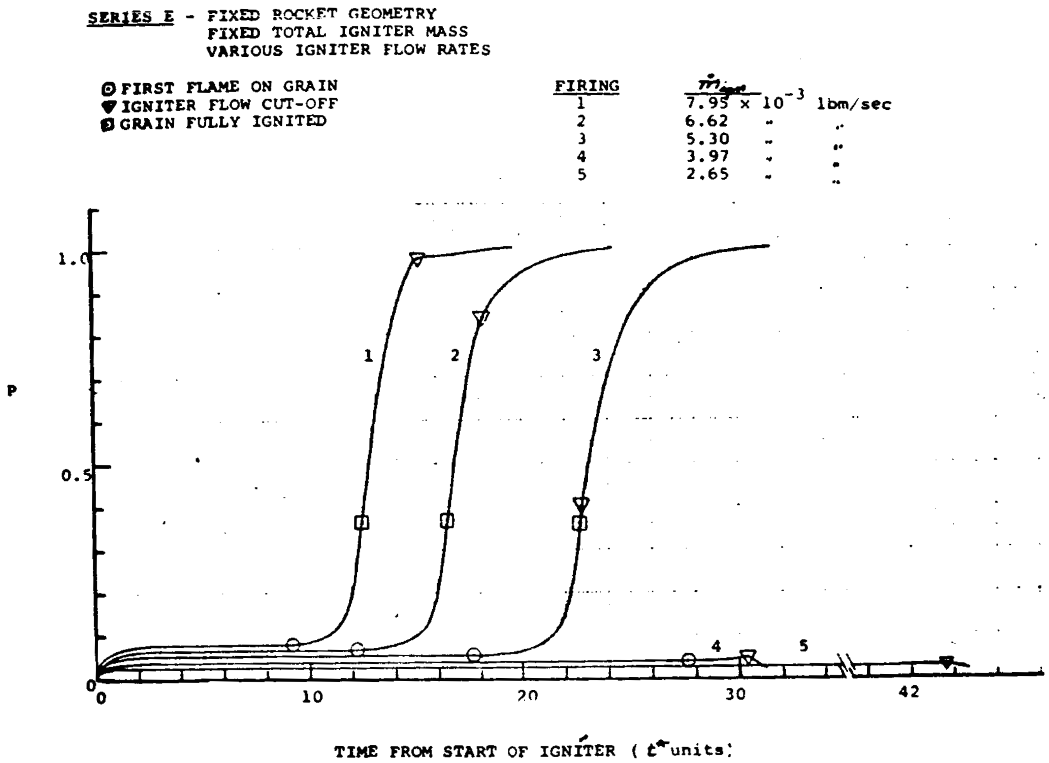


Figure 13 Theoretically Predicted Effect of Varying Igniter Flow Rate with Total Igniter Mass Constant

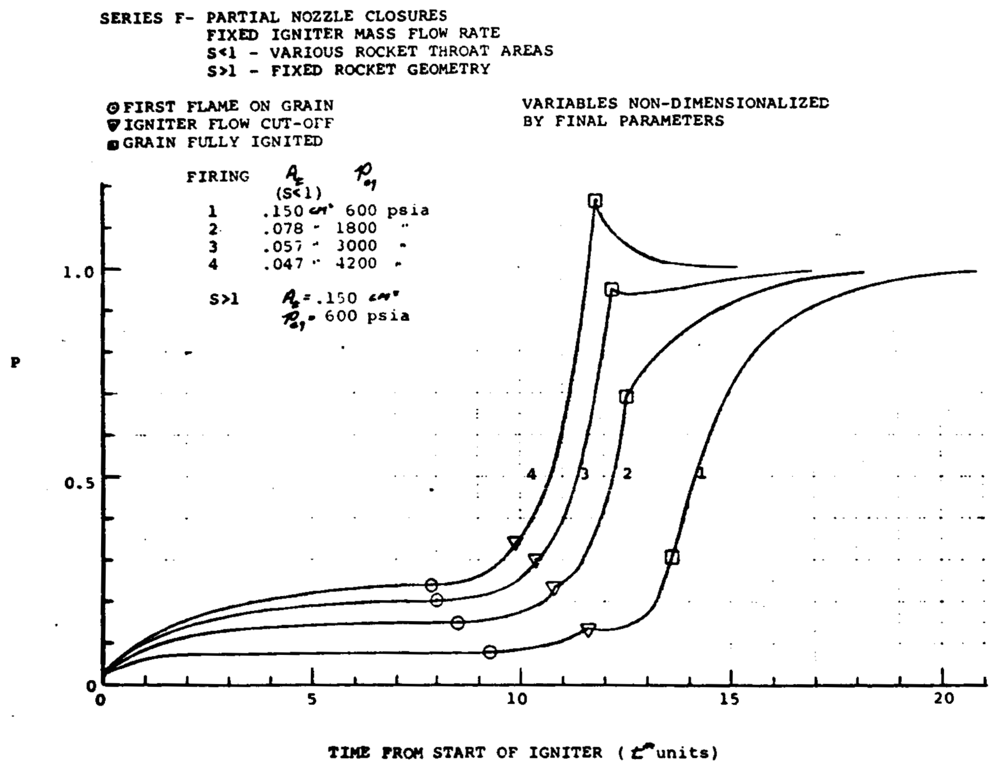


Figure 14 Theoretically Predicted Effect of Frangible Partial Nozzle Closures